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ADP010648

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TREATMENT OF HIGH-CYCLE VIBRATORY STRESS IN ROTORCRAFT DAMAGE TOLERANCE DESIGN

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SUMMARY

Fixed wing aircraft manufacturers have adopted the damage tolerance design philosophy with great success for both military and commercial aircraft. However, rotorcraft manufacturers currently still primarily use the classical safe life approach or a modification thereof. One reason for this is that, at this time, no clearly defined damage tolerance design criteria exist for rotorcraft structures because of the analysis and test problems associated with the high cycle loading environment.

This paper describes a study performed by the United States Air Force (USAF) to assess the impact of the damage tolerance approach on the design of a rotorcraft component affected by high-cycle vibratory stresses. The assessment consisted of developing the stress spectrum for a critical rotor system location and performing fracture analyses to determine the potential for establishing inspection intervals based on the damage tolerance approach. They performed sensitivity studies to determine the maximum range truncation that would yield results with acceptable accuracy. They considered the influence of the small-crack effect in all fracture mechanics calculations. The resulting crack growth functions provided the basis for establishing whether an inspection program was viable for the component. They examined the effect of stress reduction measures such as shot peening to enhance the damage tolerance capability of highly stressed components. Therefore, the paper identifies the main issues related to the use of damage tolerance for rotorcraft, and additionally makes recommendations for rotorcraft design criteria.

INTRODUCTION

The USAF made the decision to adopt damage tolerance in the early seventies. They did this when it became apparent that the safe life approach adopted through the Aircraft Structural Integrity Program (ASIP) in 1958 in response to fatigue failures in operational aircraft (1) did not achieve the desired results. These service failures resulted in costly redesign and modification programs on many aircraft. The USAF, in part, attributes the failure of the safe life approach to the fact that it did not properly account for the possibility of a large remotely occurring or "rogue" defect in the structure. The manufacturing process or in-service maintenance of the aircraft may induce this type of defect. Now, all of the major weapons in the USAF have had their Force Structural Maintenance Plans (i.e. the how, when and where to inspect or modify an aircraft to maintain safe and economical operations) updated by the damage tolerance approach.

The USAF believed that the damage tolerance approach they developed for fixed wing aircraft (2) could apply to engine structures. They have done this with considerable success. Initially they applied the damage tolerance approach to the TF-41. Subsequently, in 1978, they applied it to the F-100 engine used in the F-15 and F-16. This assessment proved to the USAF that if they had incorporated damage tolerance methods in the initial design of this engine, the damage tolerance requirements would have precluded the use of the low toughness materials. In addition, the manufacturer would have been able to better balance the design for durability. Retirement for cause (i.e. the damage tolerance approach) has demonstrated a substantial reduction in the maintenance burden and improved safety of the parts that the USAF previously managed under the safe life concept. Later, damage tolerance applications to other engines also demonstrated that this approach was technically sound.

Based on success with fixed wing aircraft and engines the United States Air Force (USAF) in 1983 sponsored a damage tolerance assessment (DTA) of the HH-53 helicopter (3). Sikorsky performed this pioneering assessment based on the approach that had proved to be successful in the earlier DTAs. The HH-53 DTA posed a considerable challenge to Sikorsky. Their approach to the design of rotor components had been a modification of the safe life approach used by the USAF. They used fatigue methodology based on Miner's rule of linear cumulative damage. They tested component parts with a constant amplitude spectrum to determine the fatigue strength. They assumed a normal distribution for the fatigue strength and used a "working" S-N function that is approximately three standard deviations from the mean. This process is similar to that used by the engine manufacturers to determine low cycle fatigue (LCF) lives for their production parts. This process did not require Sikorsky to perform a detailed stress analysis of the components tested. It did incorporate, however, conservatively derived data from flight tests to determine the LCF lives. The damage tolerance approach, however, does require that the analyst accurately know the stress spectrum. The stress analysis and the determination of the usage were major

Paper presented at the RTO AVT Specialists' Meeting on "Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft", held in Corfu, Greece, 21-22 April 1999, and published in RTO MP-24.

efforts in the DTA. Sikorsky, however, could not remove all of the conservatism in the applied loads for the DTA because of time and budget constraints.

In recent years, researchers have found a connection between the results of a classical safe life testing and fracture mechanics. R. Everett, in his comparison of fatigue life prediction methodologies for rotorcraft (4), found that constant amplitude testing of polished coupons and the crack growth method based on "small-crack" crack growth data and a crack-closure model yielded comparative results. He performed this study on AISI 4340 steel. This is a material of choice for many rotor components. Based on earlier work he used an initial flaw size of 15 microns. This size defect is approximately an order of magnitude smaller than could be found by currently available inspection methods.

At the time of the initiation of the HH-53 DTA the USAF had accumulated approximately one million man-hours of experience with fixed wing aircraft DTAs and approximately 200,000 man-hours of experience with the engine assessments. Therefore, it was logical to use this background as a pattern for the helicopter effort. The process (5) included the following five tasks:

(1) Identification of critical areas

(2) Development of the stress spectrum

(3) Establishment of the initial flaw criterion

(4) Establishment of the operational limits

(5) Development of the Force Structural Maintenance Plan

The critical areas are those where an inspection or modification is required during the life of the structure. As was found for fixed wing aircraft, the analyst augments the identification of critical areas from knowledge of the service problems, laboratory fatigue testing, stress analyses, and surveys for strain. There are, however, some major differences between the fixed and rotary wing aircraft. The laboratory fatigue testing is concentrated on the rotor systems and in many cases, there is no full-scale fatigue test on the helicopter airframe. In addition, the manufacturer performs most of the laboratory testing of rotors on a component basis to support the safe life analysis and consequently there is a question whether they have identified all areas of structural concern. Lack of comprehensive full-scale fatigue testing in combination with a lack of a detailed stress analysis and strain survey data compounds this problem. These problems force the analyst to initially identify the critical areas in somewhat conservative fashion and then reduce the number for a detailed examination by a preliminary screening process. Typically, this initial identification would include, as a minimum, all of the components in the main and auxiliary rotor systems and all of the airframe components influenced by the rotor system dynamics. This preliminary screening activity usually involves finite element analyses of some components followed by a simplified evaluation of the stress spectra and a fracture analysis.

The stress spectrum development was typically the most difficult to perform of all the tasks for the fixed wing DTAs. Much of the difficulty arises because of the need for accuracy. This need for accuracy is also vital for the helicopter. As stated in (2) there are three phases in the development of the stress spectrum for a fixed wing aircraft. For a helicopter, the procedure is similar for the airframe and rotor systems. For both the airframe and rotor systems, the desirable first phase is to derive operational loads and stress data from service usage. However, helicopter usage tracking data is essentially nonexistent. Therefore, to develop an estimate of the usage the analyst will need to conduct interviews with pilots. This, in combination with knowledge of the basic mission of the vehicle and a general knowledge of how helicopters are used will provide a usable basis until the operators initiate an individual aircraft-tracking program. The second phase in the development of the stress spectrum for the airframe is to determine the external loads on the structure for a given maneuver or gust condition. This process is generally tractable with analytical procedures for the external loads on the airframe. The exception is the loads derived from the dynamics of the rotor systems. For rotor locations and airframe locations that have dynamic loads from the rotor system imposed on them, the manufacturer generally obtains the external loads by direct strain measurements from these locations during flight test. The third phase of the stress spectra development is to determine the detail stresses in the critical areas resulting from the external loads. The USAF found in the entire fixed wing DTAs that knowledge of the detail stresses was lacking. This was due, at least in part, to lack of computer capability at the time of the acquisition process and use of the safe life method for qualification. Consequently, the DTA needed to have a significant number of man-hours expended to correct this deficiency. This situation also exists for helicopters.

The desired product of the stress spectra development is a flight-by-flight spectrum of stresses for each of the critical areas. In the DTAs of fixed wing aircraft, the USAF found that a change in the ordering of the flights in the spectrum does not have a significant effect if they used a random ordering. They found this true also for the helicopter stress spectra. In addition to the development of the baseline spectra, which they intended to represent average usage, they found it necessary to generate variations of these spectra that represent the expected range of usage of the vehicle. The successful completion of the fracture analyses and

tests with these spectra provides confidence that there is a sound basis for the tracking program.

The third task of the DTA is for the analyst to determine the initial flaw size for the subsequent fracture analysis. The USAF experience, derived from the fixed wing DTAs, has shown that for fastener holes in airframe structure an initial flaw of 1.27 mm is adequate to provide safety of flight. They derived this initial flaw size from the observation that they have never experienced an aircraft failure in the flight time required for this size flaw to grow to critical. They have also found that for airframe structure that a semicircular surface flaw with a radius of 3.175 mm is similarly adequate. Although the initial flaw basis for helicopter airframe structure is limited, the flaw sizes above should be adequate since the manufacturing processes are similar to fixed wing aircraft. The USAF does not believe, however, that these flaw sizes are the minimum for the rotor structure components since the quality control of these components is better than for airframe components. The USAF believes that the rotor DTA should use the engine DTA approach for establishing the initial flaw size. For DTAs performed on engines, the USAF approach is quite different than for those performed on airframe structure. For engines, they perform a deterministic DTA for surface flaws with initial flaw sizes based on inspection capability. For imbedded defects, the approach is probabilistic based on perceived distribution of initial internal defects. Experience with engine components has shown that smaller initial flaw sizes than used for airframe are adequate to provide safety. For engines, the USAF established the surface flaw size for initial design (5) as 0.762 mm, and they established the corner flaw the initial flaw size as 0.381 mm. The USAF recommends that manufacturers use these flaw sizes as the initial defects of helicopter rotor components. As was found for the engine assessments, the inspection capability for helicopter rotor components is extremely important. The USAF found that inspection capability was the essential ingredient that made retirement for cause for engines a viable concept. The Engine Structural Integrity Program (ENSIP) Military Standard (5) released in 1984 states that the detectable flaw size using eddy current or ultrasonic surface wave techniques is 0.381 mm (uncovered length). This capability is adequate for many of the helicopter rotor components. It is a practical limit since smaller cracks introduce problems in testing for crack growth and the "short crack" phenomenon emerges to complicate the fracture process. Further, and possibly a more important consideration, is that when a structure is sensitive to flaws smaller than this, the manufacturer should modify the structure rather than inspect it to maintain its safety for service operations.

The analyst derives the operational limits for the helicopter from the fracture analyses and tests based on the stress spectrum for each critical area in combination with the initial flaw associated with that area. As reported in (2), the predominant problem with fixed wing aircraft is fastener holes. This is, in general, not the case with the rotor system critical areas. In many cases, such as threaded parts and lugs, the stress intensity solution is more difficult to determine. However, in rotor systems it is rare to find that the critical crack size is large enough to change the stress distribution from that found when the structure is in its pristine condition. As with airframe structures, testing is essential for verification of crack growth for rotor system structure. These tests are difficult because of the large number of cycles in the stress spectrum for some parts.

The Force Structural Maintenance Plan (FSMP), as indicated above, is the plan that describes inspection and modification program for the helicopter during its anticipated operational life. The analyst determines the inspection intervals from the operational or safety limits (i.e. the crack growth life from an initial or detectable flaw to critical crack size). The USAF recommends dividing the safety limit by a factor of two to determine the inspection interval. This procedure is identical to that used for aircraft with fixed wings.

The results of the HH-53 DTA showed that many of the rotor component areas are damage tolerant. That is, an inspection program could be developed that would permit economically viable inspection periods. There are other areas, however, that are subjected to such a severe environment that the time to critical size was unacceptably short even from an initial flaw of 0.127 mm. These locations would be candidates for modification. These large differences would probably not have been found if the manufacturer had applied the damage tolerance approach during the initial design. It is noted that the component replacement time for the spindle threads in the tail rotor, which has a 10 hour crack propagation time from a 0.127 mm deep flaw, has a recommended component replacement time of 11,000 hours. There are cases where the recommended component replacement time is less than the crack propagation time from a flaw of 0.127 mm. Thus, the DTA revealed a lack of consistency between the recommended component replacement times and the crack propagation times. This paper discusses some possible reasons for these problems and suggests some strategies for solution.

DISCUSSION

There has been considerable progress made since the HH-53 DTA in the development of technologies that would enhance helicopter structural integrity. One of these is a better understanding of the "small-crack' effect. This has increased the confidence in calculations that involve crack growth in the ΔK threshold region. It has permitted the analyst to perform fracture mechanics calculations that correlate with the growth of intrinsic defects in materials such as aluminum and titanium. Figure 1 shows derived

intrinsic flaw distribution for Ti-6Al-4V developed from constant amplitude testing (6) during the F-15 development program. Another development is computer controlled shot peening. The USAF believes this process significantly enhances the reliability of shot peened parts that the manufacturer uses in safety of flight critical locations. This development is important since the integrity of many helicopter parts relies on shot peening. Another development is the technology for usage tracking of helicopter components. It has the potential for reducing the economic burden of inspection or part replacement without compromising flight safety.

There have been important developments in the certification requirements for helicopters. The Federal Aviation Administration (FAA) has made important changes through Amendment 28 to FAR Part 29. The FAA asserts in this document that the manufacturer will design a new helicopter using damage tolerance or flaw tolerance methods. They will not accept the classical safe life approach except for areas where damage tolerance or flaw tolerance is not feasible. The damage tolerance approach in these requirements is essentially the same as outlined above for the HH-53. The flaw tolerance approach is new. This method is a modification of the classical safe life method. The modification is that the parts prior to testing are subjected to an array of damages including nicks, scratches, gouges, and corrosion damage that would be representative of in-service operation experience. If the analyst maintains all of the other elements of the safe life method then this approach appears to add more conservatism to the determination of the component lives. One difficulty with this approach is convincing the authorities that the damage inflicted is representative of actual in-service damage. Another problem is that it has the potential for the same errors that are inherent with the classical safe life method. In addition, the added conservatism adds to the already significant economic burden of the usual safe life approach. The Technical Oversight Group for Aging Aircraft (TOGAA), an advisory group to the FAA, has campaigned vigorously in opposition to the flaw tolerance approach. The helicopter industry in the United States, however, does not speak with a single voice on the certification issue. Some believe that the FAA should retain flaw tolerance as acceptable approach until the damage tolerance approach has matured further. Others believe that flaw tolerance should be the only acceptable approach. These certification issues will likely not be resolved soon.

The U.S. Army and U.S. Navy develop all of the military helicopters in the United States. At this time, the certification base used by both of these services for helicopters is the classical safe life method.

EXAMPLE PROBLEM

The USAF developed a sample problem to reveal many of the difficulties associated with analyses involving high cycle fatigue. The selected sample problem is typical of helicopter component. The more obvious difficulties with an analysis such as this are as follows:

> The large number of cycles increases the computer run time significantly.
>
> Many cycles occur at relatively high mean stress.
>
> Lack of data around the crack growth threshold.

Another problem is the modification of the stress intensity solution to account for the effect of life enhancement measures such as shot peening. For this paper, the USAF used the expected change in the stress distribution as the basis for the modification. Since the high-cycle vibratory stresses accounted for a large percentage of the total cycles, the USAF investigated the effects of high-cycle stress range truncation. In the block spectrum consisting of 1000 hours of usage, the truncation greatly affected the total number of cycles. The following results clearly illustrate this for the Ti-6Al-4V material.

The analyst based the cycle-by-cycle, flight-by-flight stress spectra on a nominal block time of 1000 hours, which represents 10 percent of one service life. From the given utilization, the analyst determined the number of flights for each mission types. He then ordered the flights randomly. However, he maintained the order of the segments within each mission type.

Since each segment in each mission type consisted of particular flight or ground operation conditions, the analyst associated the appropriate low-cycle exceedances with each segment. For each segment in each mission type, he adjusted the low-cycle occurrences to account for segment time and total missions for the mission type. He then distributed these occurrences randomly across the same segments in all the flights of each mission type. Finally, he constructed a low-cycle flight-by-flight stress spectrum by randomly ordering the occurrences in each segment in each random flight.

To construct the high-cycle spectrum, the analyst superimposed the oscillatory stress cycles on each peak and valley of the low-cycle spectrum. He determined the number of high-cycles for each peak and valley from the rotor rotational frequency and maneuver time. Figure 2 shows the method of superimposing the oscillatory stresses. The addition of these oscillatory stresses vastly increased the number of cycles imposed on the structure since he superimposed up to 200 oscillatory cycles on each low-cycle peak and valley.

The damage tolerance analysis procedure is based on Forman's crack growth rate equation and requires

appropriate input data such as stress intensity factor equation, retardation factor, and ΔK threshold. The sensitivity studies indicate the effect of ΔK threshold on damage tolerance life can be significant.

Figures 3 shows the geometric configuration and Figure 4 show the stress exceedance functions. As shown in Figure 5, the high-cycle stresses reduce the life from 4500 hours to 100 hours for an initial flaw of .254mm. For an initial flaw of 1.27mm, the reduction is from 1000 hours to 50 hours. This demonstrates the oscillatory stresses have a major influence on the damage tolerance derived inspection intervals. Figure 6 shows the affect of the ΔK threshold for the high cycle spectrum. The initial flaw for the crack growth functions in Figure 6 is 0.254 mm.

To develop a practical analysis spectrum, the analyst performed a study to determine the best compromise for stress range truncation. Figure 7 shows the effect of high-cycle range truncation for an initial flaw of 1.27 mm. Below 34.48MPa, they found no noticeable difference in the crack growth function. Truncation to 51.72MPa resulted in the best balance between analysis run time and accuracy for this material. This type of study needs to be performed for each material and spectrum associated with the critical components of the helicopter rotor system. The analyses must be supported by tests with initial cracks approximately the inspectable crack size.

Truncation MPa	Cycles per 1000 hours
Low-Cycle Only	144,682
86.19	388,849
68.95	1,725,650
51.72	14,530,800
34.48	40,791,800
0.00	41,404,000

The remaining analyses for the example problem investigated the effects of shot peening and the influence of changing the stress spectrum. The analyst approximated the shot peening effect by superposition of the stress intensities for the basic geometry and stresses and the stress distribution derived from shot peening. He found the effect on life from shot peening to be very significant. For an initial defect on the high side of the intrinsic defect distribution in the material (0.127 mm), the increase in life from shot peening is of the order of 10 as shown in Figure 8. One observes for the stress used in this analysis the crack growth life from a detectable flaw is too short for a damage tolerance derived inspection program. The analyst needs to reduce the stress by a factor of 1.6 to attain an inspection interval that would be consistent with rotors with normal maintenance periods.

CONCLUSIONS

The results derived from the HH-53 DTA indicate that many of the parts in its rotor systems are critical and will require either inspections, modifications or replacement during their operational lives. Based on experience with DTAs on fixed wing aircraft and engines the USAF expected this result. In most of the DTAs on these systems, they identified many extremely critical parts. In many cases, these critical areas identified themselves through in-service failures or full-scale fatigue test failures. As indicated in (2), the USAF addressed these problems individually and the USAF determined an approach that preserved safety at minimum cost.

The indication of part criticality from the HH-53 DTA may be overly conservative since Sikorsky did not include the beneficial effects of shot peening in the calculations. The results shown in the example problem indicate that shot peening can significantly extend the inspection intervals for components with flaws that the USAF could detect with confidence. The effect of shot peening is even more dramatic for flaw growth from the intrinsic defects found in aluminum and titanium. The example problem illustrates the results that a manufacturer would derive from testing smooth fatigue specimens with and without shot peening. From this illustration, it becomes clear why there could be a large difference between the safe life calculated for a shot peened component and the inspection interval derived from a fracture analysis of the same component without shot peening. The omission of the effects of shot peening in the determination of the inspection intervals of the components of the HH-53 led to results that made the maintenance of this rotor system unmanageable through damage tolerance principles.

One observes another significant feature from the example problem. Many components in operational rotor systems have stresses so high they must rely on some life enhancement procedure such as shot peening, laser peening, or cold expansion. Although these processes enhance the life of the part, it may still be in category of damage tolerant non-inspectable. As indicated above it appears justified to use engine inspection procedures and associated inspectable flaw sizes for DTA of helicopter rotor components. In addition, as indicated above, there was some conservatism in the stress spectra used for the HH-53 assessment. The results of the DTA, however, clearly show that there is a requirement for a high reliability nondestructive inspections for small flaw sizes if the operator uses the damage tolerance approach for operational helicopters. A reasonable goal for inspection capability is a flaw size of 0.254 mm for a disassembled rotor system. The USAF believes that this capability is attainable if they inspect the parts with the techniques developed for the engine community. Further, this defect size is compatible with current methods for fracture mechanics testing for analysis

verification. This test verification of the analytical crack growth predictions is an essential ingredient of the damage tolerance process. Components that exhibit crack growth periods from this size defect less than a factor of two times a reasonable rotor disassembly time should be candidates for redesign and replacement. For the HH-53, there were several members of the structure in this category.

The damage tolerance approach generally demands that the manufacturer reduces the stresses from that required for a safe life design. The manufacturer can mitigate some of the weight impact associated with this stress reduction by the balanced design they would achieve with damage tolerance. There have been a few studies of the weight impact associated with the incorporation of a damage tolerant design. R. Boorla (7) accomplished one of these. He performed an analysis of parts (including steel, titanium, and aluminum) from the prototype of the V-22 aircraft, which was designated the XV-15. He found that the weight impact was not significant in this particular case.

The results of the sample problem study indicate that many influences impact the damage tolerance life. These include high-cycle vibratory stresses, high-cycle stress range truncation, ΔK threshold, and initial flaw size. Using the generally recommended initial flaw sizes for fixed wing airframe structures may be too severe for many rotary wing high-cycle vibratory stress components. However, use of reduced flaw sizes for analysis would require the substantiation of production and maintenance inspection methods. This study concludes that one can use the damage tolerance design approach for high cycle components if the designer considers high cycle vibratory stresses, ΔK thresholds, and appropriate initial flaw sizes in the design processes.

Finally, there is a need to generate usage data through tail number tracking systems. The use of tracking data will enable the designer to quantify the range of aircraft usage as represented by the total pilot population rather than the experienced flight test pilot. Further, it will permit the maintenance organization to recognize when the helicopters have undergone a usage change and modify their maintenance actions accordingly. The technology is currently available to perform the tracking function.

ACKNOWLEDGEMENT

The authors express their gratitude to Mr. Robert Gerami and Mr. Doug Cornog of USAF Aeronautical Systems Center for their development work on the high cycle spectrum.

REFERENCES

1. Negaard, G. R., 1980 "The History of the Aircraft Structural Integrity Program," Aerospace Structures Information and Analysis Center (ASIAC) Report No. 680.1B.

2. Lincoln, J. W., 1985 "Damage Tolerance - USAF Experience," Proceeding of the 13th Symposium of the International Committee on Aeronautical Fatigue, Pisa, Italy.

3. Lincoln, J. W., 1987 "Damage Tolerance for Helicopters," Proceeding of the 14th Symposium of the International Committee on Aeronautical Fatigue, Jerusalem, Israel.

4. Everett, R. A. Jr., 1992" A Comparison of Fatigue Life Prediction Methodologies for Rotorcraft" Journal of the American Helicopter Society

5. Department of the Air Force, 1984 "Engine Structural Integrity Program (ENSIP)," Military Standard MIL-STD 1783.

6. Lincoln, J.W. and Melliere, R.A., 1998 "Economic Life Determination for a Military Aircraft," AIAA/ASME/ASCE/AHS/ASC Structures, Dynamics, and Materials Conference, Long Beach, CA.

7. Boorla, R., 1984 "Damage Tolerance Trade-Off Study," Bell Boeing Report No. 901-910-001.

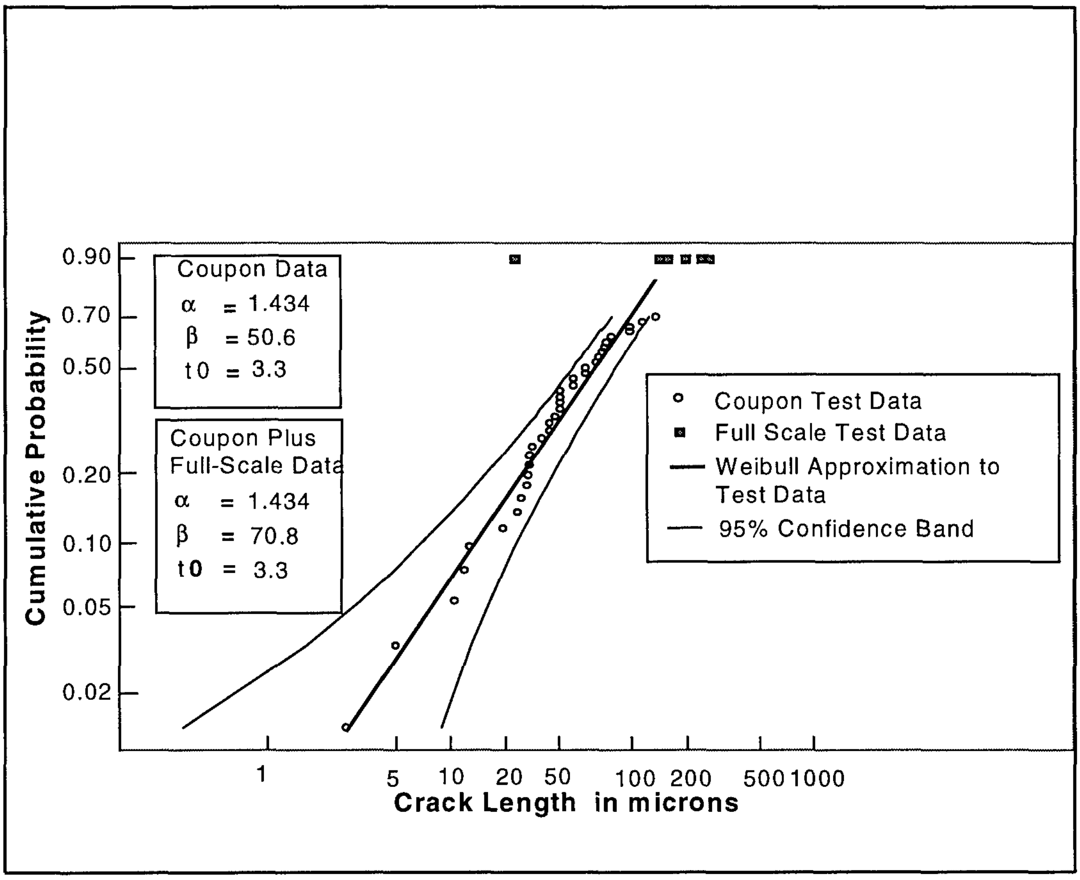

Figure 1 Intrinsic Flaw Distribution

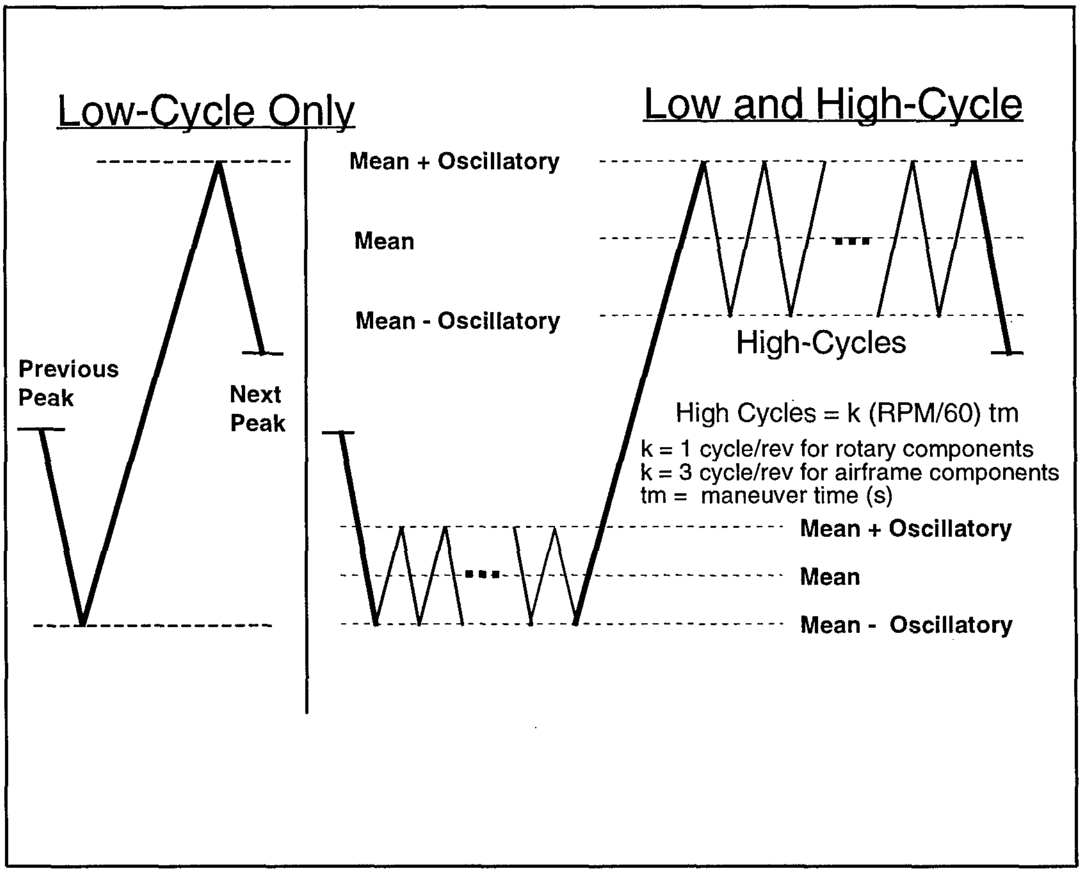

Figure 2 Addition Of High-cycle Stresses

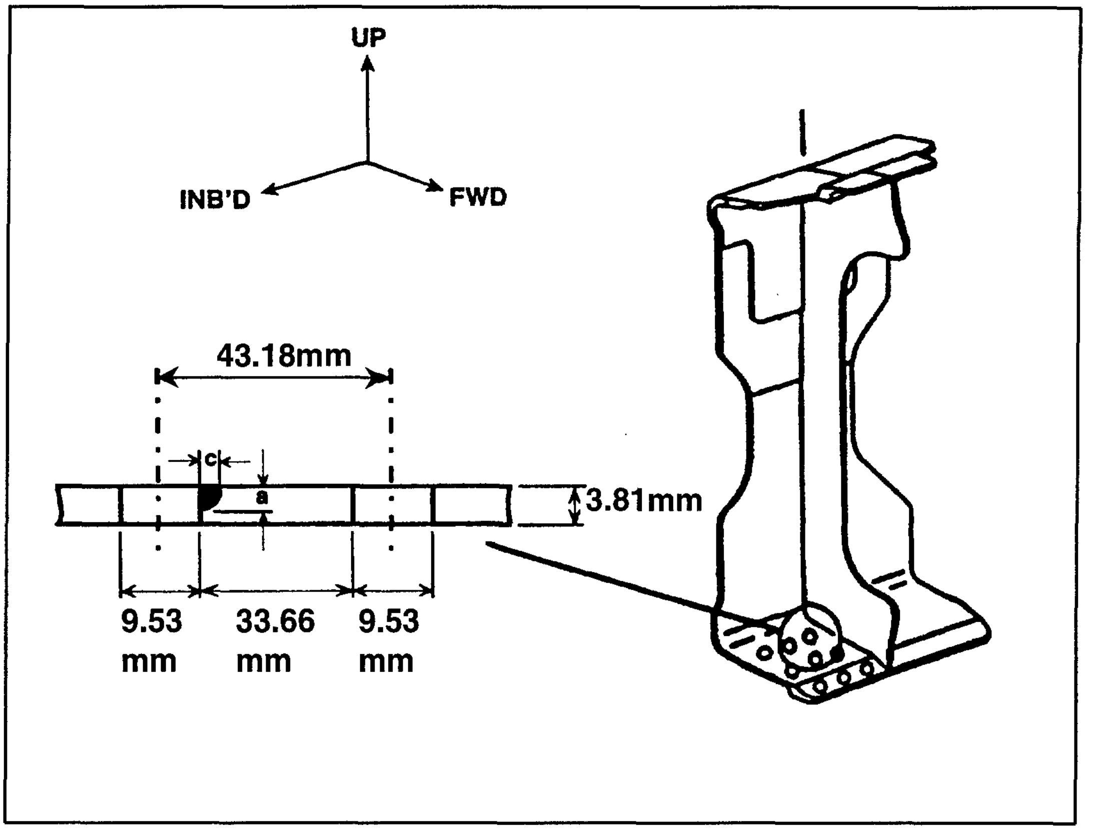

Figure 3 Splice Fitting Fastener Hole

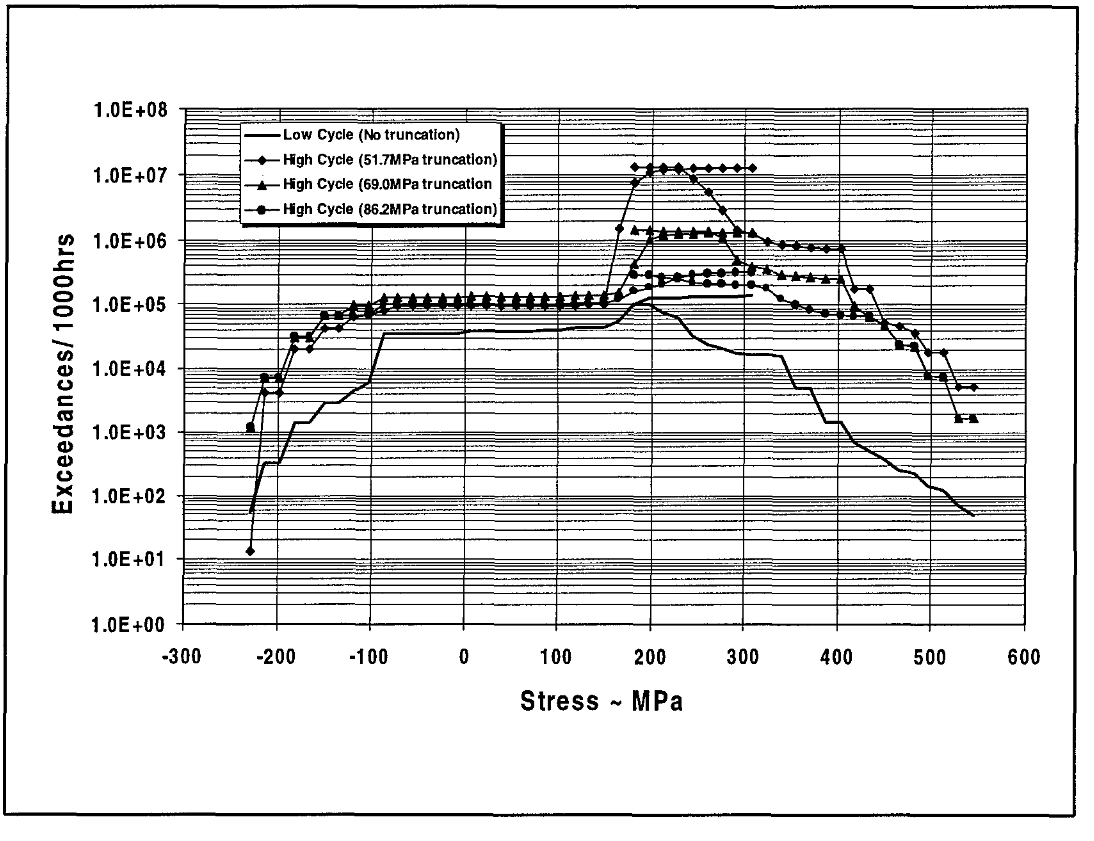

Figure 4 Stress Exceedance Functions

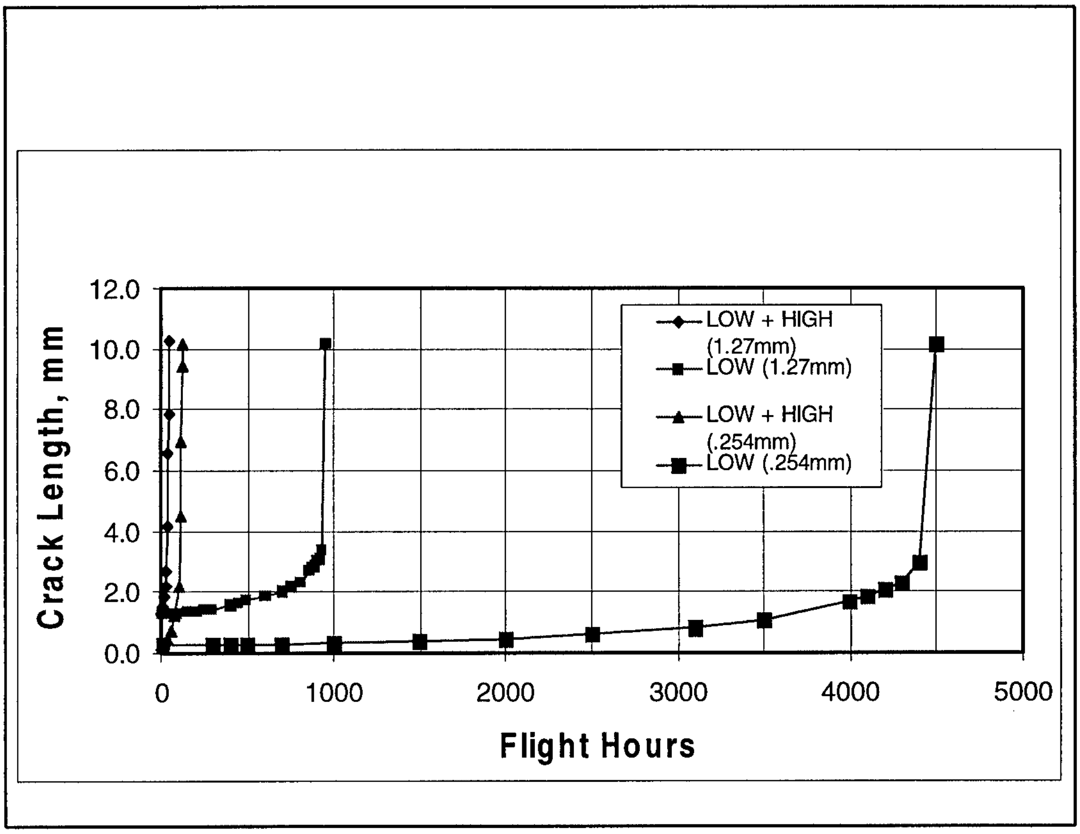

Figure 5 Structure Damage Tolerance Life

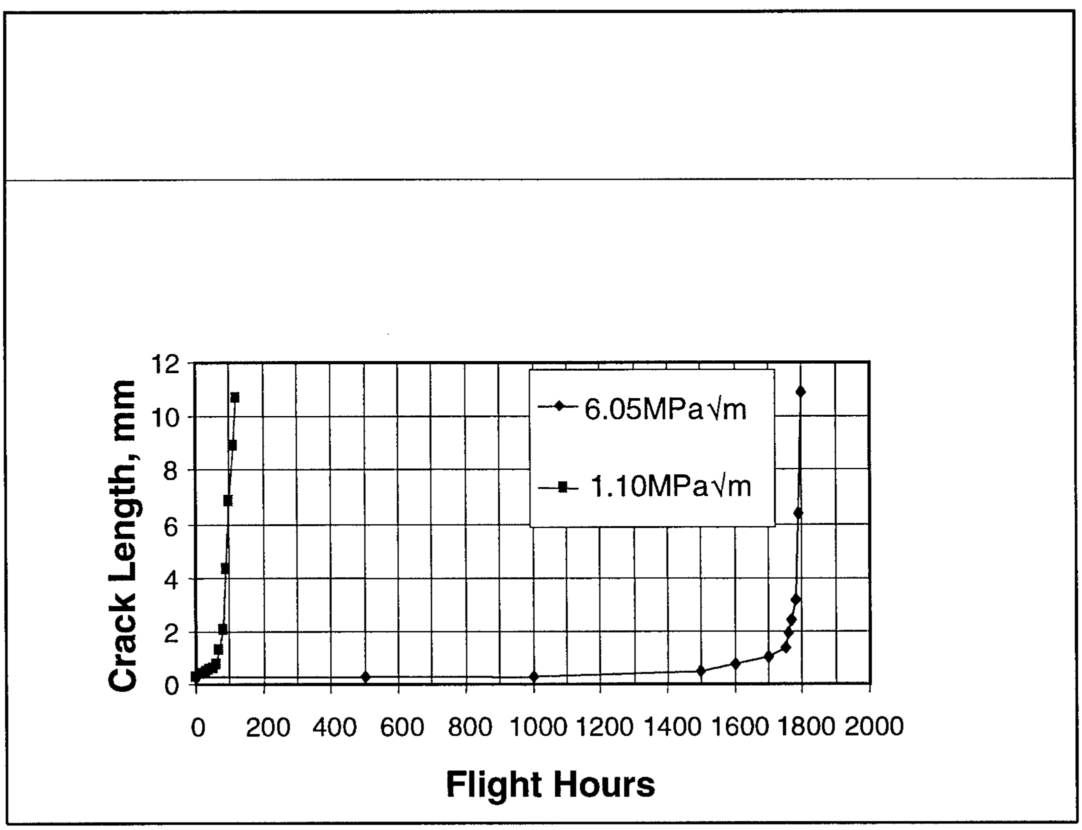

Figure 6 Effect Of ΔK Threshold On Structural Life

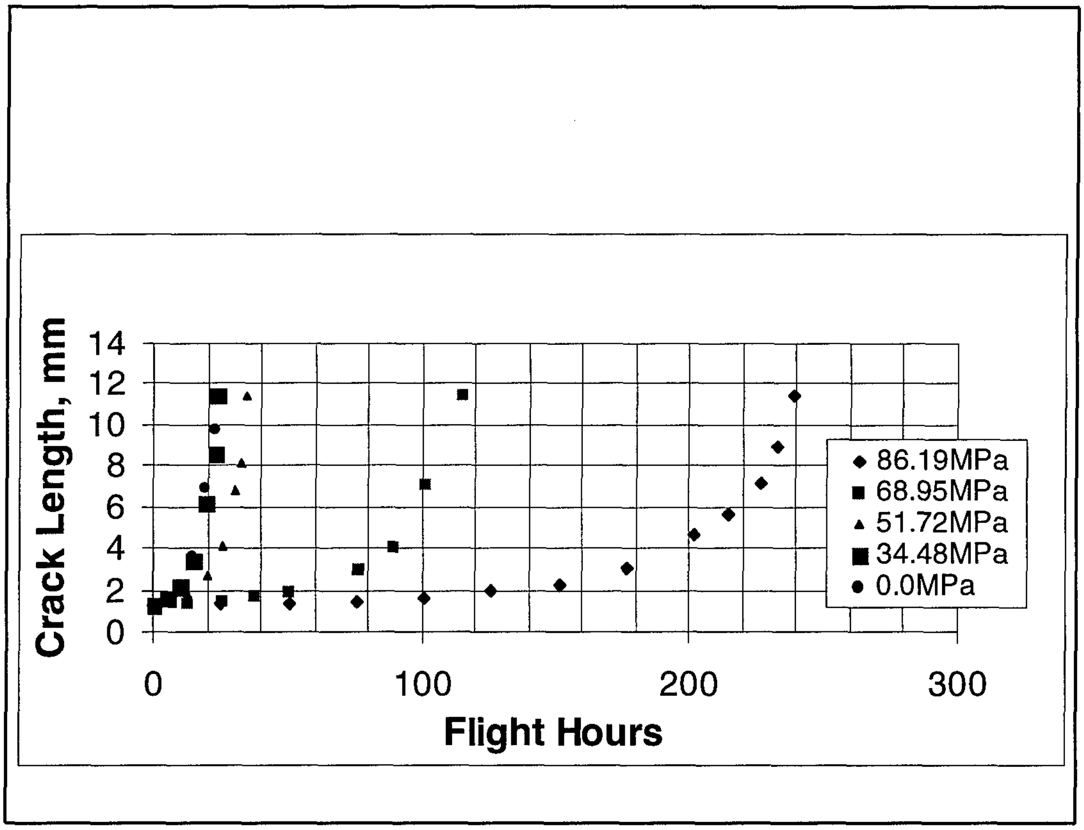

Figure 7 Effects Of Stress Range Truncation

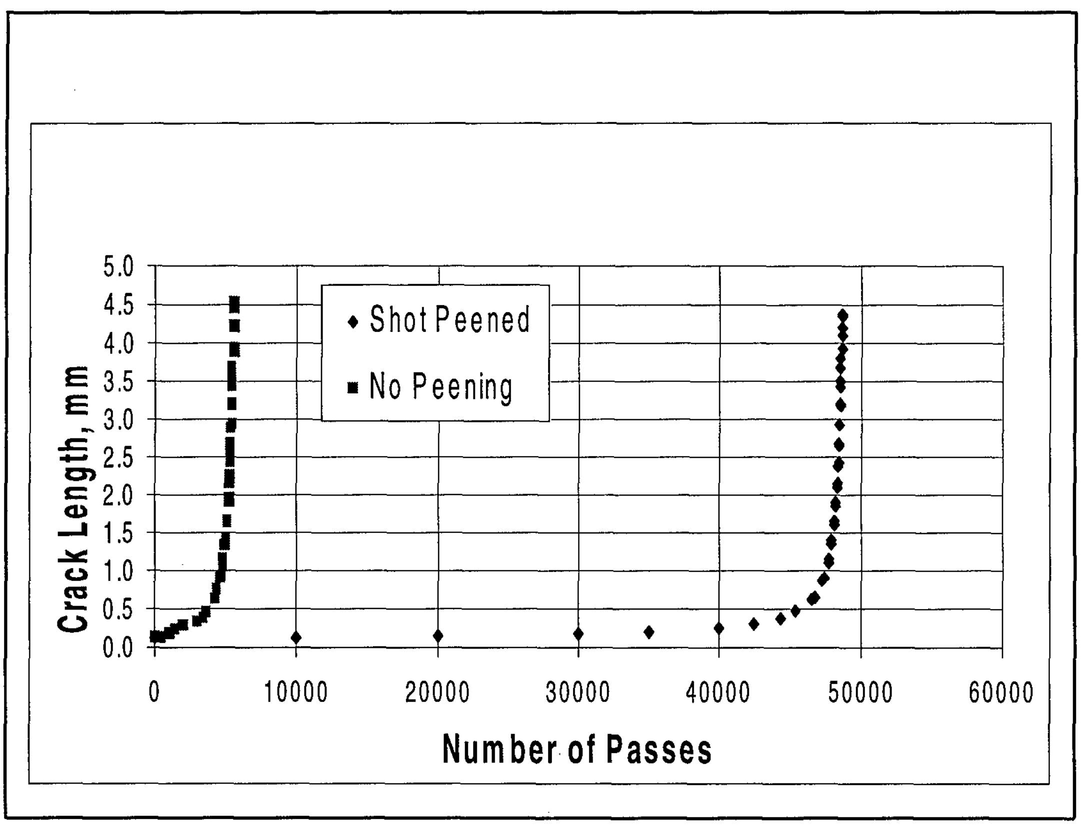

Figure 8 Shot Peening Study